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Ames Research Center, Moffett Field, California

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Moffett Field, California 94035-1000

TECHNOLOGIES FOR AEROBRAKING

David M. Cooper* and James O. Arnold†
NASA Ames Research Center
Moffett Field, California

Abstract

Aerobraking is one of the largest contributors to making both lunar and Mars missions affordable. The use of aerobraking/aeroassist over all-propulsive approaches saves as much as 60% of the initial mass required in low-Earth orbit (LEO), thus reducing the number and size of Earth-to-orbit launch vehicles. Lunar transfer vehicles (LTV), which will be used to transport personnel and materials from LEO to lunar outposts, will aerobrake into Earth's atmosphere at approximately 11 km/sec on return from the lunar surface. Current plans for both manned and robotic missions to Mars use aerocapture during arrival at Mars and at Earth return. At Mars, the entry velocities will range from about 6 to 9.5 km/sec, and at Earth the return velocity will be about 12.5 to 14 km/sec. These entry velocities depend on trajectories, flight dates, and mission scenarios and bound the range of velocities required for the current studies. In order to successfully design aerobrakes to withstand the aerodynamic forces and heating associated with these entry velocities, as well as to make them efficient, several critical technologies must be developed. These are vehicle concepts and configurations, aerothermodynamics, thermal protection system materials, and guidance, navigation, and control systems. This paper describes the status of each of these technologies and outlines what must be accomplished in each area to meet the requirements of the Space Exploration Initiative.

Nomenclature

A	reference area, m^2
AFE	aeroassisted flight experiment
ASTV	aeroassisted space transfer vehicle
C_D	drag coefficient, dimensionless
CFBI	composite flexible blanket insulation
ECCV	Earth crew capture vehicle
GN&C	guidance, navigation and control
IMLEO	initial mass [placed in] low-Earth orbit
I_{sp}	specific impulse, sec
L/D	lift-to-drag ratio
LEO	low-Earth orbit
LEV	lunar excursion vehicle
LLO	low-lunar orbit
LTV	lunar transfer vehicle
m	vehicle mass, kg
MEV	Mars excursion vehicle
MTV	Mars transfer vehicle
NTR	nuclear thermal rockets
OEX	orbiter experiment
SEI	space exploration initiative
TABI	tailorable advanced blanket insulation
TPS	thermal protection system
γ	entry corridor width

Introduction

NASA's current plans for the future exploration of the Solar System will focus on returning to the Moon and then on human missions to Mars. Precursor robotic missions to Mars are an integral part of the current plan. The time frame as well as the success of these missions will depend on maximizing the payload in low-Earth orbit (LEO), on the planetary/lunar surface, and on return to Earth, all of which are limited by cost and current launch capability. A key enabling technology to reduce propellant mass requirements is the development and use of aerobraking techniques in which aerodynamic forces rather

*Chief, Computational Chemistry Branch. Member, AIAA.
†Chief, Thermosciences Division. Associate Fellow, AIAA.

than retropropulsion are used to decelerate for orbit changes (transition from a transfer trajectory to a planetary orbit) at Mars as well as upon return to Earth from either the lunar surface or Mars.

Aerobraking results in typical mass savings in LEO of 20 to 60%, depending on such factors as destination and mission scenario.[1-11] A typical lunar mission profile for delivering crew and cargo to the lunar surface is illustrated in Fig. 1 (Ref. 12). A piloted mission delivers a crew of four and cargo to the lunar surface and returns a crew of four and limited cargo to Space Station Freedom using an aerocapture maneuver. The results of a detailed study to compare the amount of initial mass placed in low-Earth orbit (IMLEO) for an all-propulsive mission with that of an aeroassisted mission for a typical lunar mission (Fig. 1) are given in Fig. 2. As shown, the saving of IMLEO for an aeroassisted mission over that of a mission using an all-propulsive LOX/LH2 system is plotted as a function of aerobrake efficiency. The aerobrake efficiency is simply defined as the ratio of the aerobrake weight (thermal protection system and structural elements) to the total weight of the return vehicle. For the data in Fig. 2, it is assumed that separate lunar transfer vehicles (LTV) and lunar excursion vehicles (LEV) would be used for the mission. Furthermore, the LEV is assumed to be already in lunar orbit and the return crew module remains attached to the LTV. The translunar injection tanks were expended after the lunar injection phase was completed, and the low-lunar-orbit (LLO) tanks were also expended after the propellent was transferred to the LEV.

For comparison with the propulsion-only scenario, the LTV was assumed to have a specific impulse (I_{sp}) of 481 sec, and the LEV was assumed to have an I_{sp} of 465 sec. Lunar LOX was assumed to be unavailable. The LTV and LEV propellant margins were 2% for flight performance reserve, 3% ullage, 1% residual, and 1% boil-off per 30-day month. The mission under consideration delivered a 27-tonne payload to the lunar surface and returned a 1-tonne payload to Earth orbit. As illustrated in Fig. 2, the break-even point for the aeroassisted missions is an aerobrake efficiency of 48%. This mass fraction in considerably more than the design goal of 15 to 20% which for a typical lunar mission would result in a savings of 16 to 20% in IMLEO. Table 1 is a summary of the results of several studies which have attempted to document the aerobrake efficiency or mass fraction of the aerobrake for lunar missions.[5] The mass fractions vary from 6.4 to 22.0% and demonstrate that a design goal of 15% is realistic and attainable.

A typical mission to Mars is illustrated in Fig. 3 (Ref. 12). There is a separate Mars transfer vehicle (MTV) and Mars excursion vehicle (MEV). These vehicles are connected during the trans-Mars injection phase; they separate upon approach to Mars and perform aerobraking maneuvers to enter the Martian atmosphere separately. The vehicles rendezvous in Mars orbit, and the crew of four transfers to the MEV which descends to the surface using the same aerobrake. After completion of the tour of duty on the surface of Mars, the crew returns to Mars orbit and transfers to the MTV which is used to transport the crew back to Earth orbit. After an aerobraking maneuver, the crew rendezvous with Space Station Freedom.

The results of a study similar to that described above for a Mars mission are given in Fig. 4 (Ref. 5). Once again the saving of IMLEO for an aeroassisted mission over a propulsion-only mission is plotted as a function of the aerobrake efficiency. In Ref. 5, a conjunction-class mission with an outbound trip time of 205 days, a 30-day stay on the surface of Mars, and a return trip time of 225 days were assumed. The MTV and MEV were assumed to have similar aerobrake efficiencies and both aerobrakes were jettisoned at Mars. In the study reported in Ref. 5, in contrast to several others in which the MTV aerobrake is used both at Mars and on return to Earth to rendezvous with Freedom, the crew transfers to an Earth crew capture vehicle (ECCV) (a small Apollo-like capsule) upon approach to Earth for a direct return to Earth's surface. For comparison, the MTV was assumed to have an I_{sp} of 481 sec and the MEV was assumed to have an I_{sp} of 465 sec. The velocity reduction for the MTV and MEV capture at Mars was assumed to be 3,400 m/sec for the all-propulsive case. The mission under consideration delivered a 25-tonne payload to the surface of Mars and returned 1 tonne to Mars orbit. Figure 4 demonstrates that the break-even point for this Mars mission is an aerobrake efficiency of 57%. An efficiency of 15%, the design goal, will according to this study provide a 50% reduction in IMLEO. As discussed earlier, a mass fraction of 15% has been demonstrated to be attainable.

Nuclear thermal rockets (NTR) have been proposed as an alternative means of transportation for the Earth-Mars exploration programs.[12] Preliminary studies suggest that Mars missions using only NTR propulsion methods with an I_{sp} of 925 sec could result in a savings in IMLEO of about 25% over an aerobrake mission based on an aerobrake efficiency of 15%. However, the shielding requirements for long-duration NTR missions is very uncertain and considerably more mass may have to be added to limit the total annual radiation dose to the crew to the currently acceptable level of 50 rem. This would of course reduce the advantage of the NTR missions over those using aerobraking. In addition, the studies discussed

above[5] have also compared the IMLEO requirements for a NTR mission with one utilizing both NTR and aero-assist techniques. The results indicated that the combined mission will result in a 10% saving (about 45 tonnes) over the all-propulsion mission.

Technology Assessment

No currently validated aerobraking capability exists. Apollo and Space Shuttle experience with simple blunt configurations operating over narrow entry corridors has yielded a very limited data base relative to aerobraking. Both of these missions used trajectories that returned directly to Earth's surface in contrast to the aerobrake maneuver, which features deceleration to orbital speed high in the planetary atmosphere. These orbits are illustrated in Fig. 5, which is a comparison of the flight regimes of selected vehicles and missions for Mars and Earth aerocapture and entry. The Aeroassisted Flight Experiment (AFE), scheduled for 1994, will demonstrate aerobraking and provide flight data for validation of critical aerobraking technologies for geosynchronous and lunar return missions.[13] These technologies are as follows: 1) vehicle concepts and configurations, 2) aerothermodynamics, 3) thermal protection system (TPS) materials, and 4) guidance, navigation, and control (GN&C).

Vehicle concepts and configurations that balance the trade-offs between aerodynamics, thermal loads, GN&C, g-loads, etc. must be developed and optimized. Aerothermodynamic characterization of the entry environment is crucial to the correct prediction of the aerodynamic forces and to the design of the aerobrake's thermal protection system (thus eliminating the extreme conservatism built into previous entry-program heat shields). The development of advanced, reusable high-temperature TPS materials is extremely important in minimizing the mass fraction of the aerobrake heat shield and thus optimizing the advantage of the aerobrake. Accurate GN&C is critical to maintaining control of the vehicle through the atmospheric encounter to provide accurate orbital conditions while maintaining structural load limits and heating constraints.

The technology requirements for the low-energy regime of aerobraking (lunar return velocities of about 11 km/sec and Mars entry velocities of less than about 7 km/sec) are not as demanding as those for the high-energy regime. Nevertheless, advances are required in these critical technologies for both the low- and the high-energy regimes of aerocapture. In the text below, the status of each of these critical technologies is reviewed. In addition, the developmental requirements necessary to advance these technologies in order to maintain aerobraking as a feasible option for space exploration missions are given.

Technology Status and Requirements

Vehicle Concepts and Configurations

The currently proposed list of missions[12] for the Space Exploration Initiative will use aerobraking to accomplish a variety of goals under vastly different operating conditions. Consequently, numerous configurations and concepts must be studied to ensure that the optimum shape is selected for each application in order to allow for proper compensation of atmospheric, navigational, and aerodynamic uncertainties. The shapes must also be designed to control the thermal heating environment and gravitational loads, as well as to meet rigid GN&C requirements. Furthermore, these configurations, particularly for Mars missions, must consider the packaging requirements and radiation protection for the crew, as applicable. Figure 6 illustrates the range of lift-to-drag (L/D) ratios for previous aerobrake vehicles, as well as those for concepts proposed for future missions. For Earth reentry missions, current studies indicate that the lower L/D vehicles (0.3 to O.5) such as the Apollo and AFE shapes are adequate, providing that the TPS requirements on both the forebody and afterbody and the GN&C requirements are met.[8-10]

For Mars entry, current studies[14-16] indicate that the L/D for the aerobrake must be O.5 or greater. Figure 7 is a plot[16] of the flyable entry corridor width (γ) versus the entry velocity for Mars aerocapture for a vehicle with an L/D of 0.5 (Ref. 16). The present acceptable minimum corridor width for successful aerocapture is 1°. It should be noted that the flyable corridor width on Fig. 7 is possible only by using an advanced predictor-corrector guidance algorithm. Without this method of guidance, the corridor width shrinks to that shown by the lower curve and would restrict the entry into Mars (for a vehicle with an L/D of 0.5) to velocities between 6.0 and 7.3 km/sec. Presently, the GN&C requirements coupled with the acceptable g-loads at Mars suggest that the Martian aerobrake have an L/D near 1.0. In addition, the cross-range requirements for landing at a permanent base, as well as the capability to explore a greater portion of the Martian surface,[17] suggests that the L/D should be even greater than 1.0.

For aerocapture vehicles using current GN&C methods, the L/D determines the width of the entry corridor, and the ballistic coefficient determines the altitude of the

corridor. (The ballistic coefficient is m/C_DA, where m is vehicle mass in kilograms, C_D is the dimensionless drag coefficient, and A is the reference area in square meters.) Figure 8 is a plot of L/D versus ballistic coefficient for typical vehicles. The figure is based on a vehicle of mass of 5,000 kg and a cross-sectional area of 14 m^2. The larger L/D vehicles result in higher ballistic coefficients and consequently in lower minimum altitudes for the aerocapture maneuver. This results in significantly larger convective and radiative heating rates to the vehicle and, consequently, additional TPS material requirements. Further studies in which each phase of the mission is considered and in which trade-offs between aerodynamic design (shape, L/D, m/C_DA), thermal heating, TPS materials, GN&C, and mission type, duration, and objectives are accounted for must be conducted before the optimum design can be selected.

Aerothermodynamics

The goal of the aerothermodynamics research is to define the flow-field environment about an aerocapture/entry vehicle and to develop validated computer codes that accurately predict this environment. The codes must be able to predict aerodynamic forces and moments and thermal heating loads for mission analysis and trajectory optimization and for the design of structural and thermal protection systems. Since aerobrakes must perform controlled maneuvers at high altitudes at velocities between 7 and 14 km/sec, this combination of high energy and low density presents a series of problems unlike those associated with previous entry vehicles. The Apollo and Space Shuttle vehicles encountered the most severe heating in the more dense regions of the atmosphere where chemical and thermal equilibrium dominated the flow. For aeroassist vehicles, finite-rate chemistry and thermal nonequilibrium conditions in which the flow field may be characterized by as many as three different particle temperatures (translational/rotational, vibrational, and electronic) dominate the flow.

The currently proposed lunar and Mars missions require that the aerothermodynamics codes be capable of covering a broad range of flow-field parameters and physics. Table 2 is a comparison of a variety of parameters for several different lunar and Mar missions with those of the AFE. (Note that all of the radiative heating predictions except those for AFE in this table are based on equilibrium theory.) As illustrated, the conditions can be very different. For example, Earth entry from the Moon and Mars (11 vs 14 km/sec) results in a factor-of-6 difference in stagnation point pressure and in a factor of at least 10 difference in the maximum heating to the surface. This means that an ablating material must be used (based on current materials) for the TPS system for the Mars missions, whereas a non-ablating reusable material can probably be used for the lunar mission. If an ablating material is used, the aerothermodynamic codes must be capable of predicting massive blowing as well as the interaction of the ablation products with the radiation. Furthermore, the lunar mission will involve air with only small amounts of ionization (<5%) whereas the Martian mission will encounter ionization levels as high as 25%. Other differences include surface catalysis and the associated heating resulting from this phenomenon, which must be taken into consideration for lunar missions, and boundary-layer transition, which must be accounted for in Mars missions. In addition to being capable of computing the flow-field properties of air, the codes must also consider Martian atmospheric gases.

At the present time, equilibrium forebody flow-field analyses are relatively advanced, but are applicable only to a limited range of simple geometries (moderate-angle sphere cones, spheres, ellipsoids, etc.). Figure 9 is a plot of the flow around an aeroassisted space transfer vehicle (ASTV) configuration traveling in air at a velocity of 4.0 km/sec (Ref. 18). The top half of the figure shows a calculation of the flow field based on a two-dimensional axisymmetric solution to the Navier-Stokes equations. The solution included seven air species and a multi-temperature model. The lower half of the figure shows a shadowgraph of an ASTV model in free flight on a ballistic range. As shown, the aerothermodynamic calculations accurately duplicate the key features of the shock structure: the shock standoff-distance, attachment at the model corner, recirculation in the afterbody, and the shock closure and expansion in the wake region. Nonequilibrium forebody analyses are currently undergoing rapid development. Afterbody and complete forebody/afterbody flow-field analyses are in an early stage of development. A recent three-dimensional calculation for the AFE shape at conditions near peak heating is shown in Fig. 10 (Ref. 19). A significant difference in vibrational and translational temperature is predicted to occur even at the lower altitudes. Such phenomena must be understood and accurately predicted in order to optiminally design aerobrake/aerocapture vehicles.

As mentioned above, the studies to date have dealt only with gases (mainly air) with small ionization levels. These must be expanded to include other planetary gases, including those with significant levels of ionization. In addition, state-of-the-art ablation-product/flow-field interaction is in a relatively elementary state of understanding. More sophisticated analyses are required for successful planetary exploration. These codes, which must accurately account for real-gas effects and finite-rate chemistry,

require as inputs chemical reaction rates; radiative transition probabilities for atoms, ions, and molecules; transport properties (viscosity, thermal conductivity, and diffusivity); rates of collisional excitation of rotational, vibrational, and electronic states of molecules; spectral line widths and shapes; and collisional cross sections for ionization, neutralization, dissociation, and energy transfer. Since the data base for these quantities is insufficient and frequently very difficult or impossible to obtain from experiments, computational chemistry techniques[20] will be used to provide the data bases for air and Martian atmospheric gases (Ref. 20). The projected data bases will include real-gas properties for both high- and low-density flows under equilibrium and nonequilibrium conditions.

Figure 11 is a plot of the electronic transition moment for the first positive system of the N_2 molecule. The theoretical calculation, which is accurate to 10%, is compared with several experimental measurements. These results demonstrate the ability of computational chemistry to greatly reduce the uncertainty that exists in the chemical, physical, and spectroscopic data base for molecules.

Experimental tests at flow conditions where there are real-gas effects (including nonequilibrium phenomena) and rarefied flow phenomena must be conducted in order to provide a data base for aerothermodynamic code validation. These tests will use ballistic ranges (typical results shown in Fig. 9), shock-tubes and shock-tunnels, arc-jets, and expansion-tubes to accurately measure flow characteristics in the flow field, as well as on and around a variety of sophisticated models. Since these facilities can duplicate only a small portion of the flight conditions, these tests are crucial to the success of the extrapolation of the aerothermodynamic codes from small-scale ground tests to flight conditions.

The Aeroassisted Flight Experiment will provide a flight data base for validation of aerothermodynamic codes for conditions near lunar return. Code development is currently in progress to predict the entire thermochemical nonequilibrium flow field, which includes the forebody region, the flow over the shoulder, and the unsteady flow in the base and wake regions. In addition, various chemical kinetic models are being developed and tested to predict the multi-temperature nonequilibrium relaxation phenomena encountered by the vehicle during its flight. The AFE instrumentation includes temperature and pressure measurement devices to define and validate the flight environment and vehicle aerodynamic characteristics.

Measurements of the total radiation and spectral distribution of the radiation in the forebody and afterbody regions will be used to evaluate the ability of the codes to predict the radiation (equilibrium and nonequilibrium), determine the chemical and thermal state of the gas, and determine the contributions of the individual gas species. Thermocouples will be used to measure heat-transfer data which will be used to determine the convective heating rates as well as the efficiency of materials and coatings selected to promote surface catalytic heating. This flight (scheduled for 1994) will serve to validate the codes for lunar missions; however, as discussed above, considerable effort must be expended to develop the aerothermodynamic codes for Mars entry and return to Earth from Mars.

Thermal Protection System Materials

The ultimate goal of the aerothermodynamics research is to develop codes that can accurately and reliably predict the aerobrake thermal environment and that can, therefore, be used to select and size the TPS materials. As described above, the weight of the TPS material and its supporting structure must be minimized to attain the maximum benefits of the aerocapture vehicle. Thus the ultimate goal of the TPS research is to provide high-temperature reusable (where appropriate) materials.

The variety of the proposed exploration missions results in vastly different heating environments and thermal requirements for each of the aerobrake vehicles (see Table 2). Consequently, the TPS requires a variety of materials. Figure 12 illustrates a part of the TPS requirements.[21] It shows a comparison of the predicted radiation equilibrium temperatures for several missions with the current TPS capabilities. The uncertainties in the temperatures result from a combination of the uncertainties in predicting the flow-field, radiative heating and catalysis effects. (Note that the equilibrium radiation temperature is the surface temperature of the tile.) It is clear that both insulative and ablative materials will be required for the proposed missions to the moon and Mars.

Figure 13 is a comparison of the maximum operating temperature for several candidate thermal protection materials.[22] The solid bars represent the current temperature limits, and the cross-hatched areas represent recent extensions of these limits which will be validated by orbiter experiment (OEX) and AFE flight tests. Therefore, lightweight reusable tiles similar to those on the Space Shuttle appear capable of handling all of the TPS requirements for lunar missions. In addition, current work on ceramic tiles suggests that these materials may be capable of withstanding temperatures as high as 3500°F. The current plans for SEI require a reusable material that lasts for seven missions. The new tiles appear capable of meeting this requirement. Indeed, if the tiles can

withstand temperatures as high as 3500°F, the TPS system for lunar missions would be reusable for considerably more than seven missions. Current studies tentatively suggest that these materials will also suffice for Mars entry missions. However, as discussed above, the thermal environment for Mars entry is uncertain and, until it is better defined, no decision can be made on whether insulative/radiative or ablative materials will be used for the TPS. Moreover, certain classes of missions to Mars in which the entry velocity exceeds about 8.5 km/sec will probably require ablative heat-shield materials over at least a portion of the vehicle.

The high velocities during Earth entry from Mars will probably necessitate ablative heat shields. Apollo-era ablative materials are the state of the art. These materials are inherently heavy and may not meet the design goal of a 15 to 20% mass fraction for the aerobrake. Consequently, the development of new lightweight ablators for use in advanced TPS systems is extremely important. The history of materials development suggests that continuation of a TPS research program should produce materials, even ablative ones, that can be reused for return from Mars.

In addition to the rigid materials described above, flexible materials such as the tailorable advanced blanket insulation (TABI) and composite flexible blanket insulation (CFBI) will be used as insulative materials on the afterbody and other portions of the vehicle where the heating is less severe. These flexible materials may also be used as the annulus for large-diameter, lightweight aerobrakes that are constructed on Earth and deployed in space. These aerobrakes would have a rigid core, but they would fold up similar to an umbrella for launch into LEO, thus eliminating the need for in-space assembly.

A variety of factors other than weight and temperature will also influence the selection of TPS materials for the aerobrake vehicles. These include damage tolerance from space and debris impact and to long-term space exposure, as well as repairability, refurbishment capability, oxidation resistance, and resistance to abrasion/impact to be encountered when flying through Martian dust storms during entry.

Guidance, Navigation, and Control

The success and performance of aerocapture vehicles depends on whether the vehicle's trajectory can be accurately controlled as it passes through the atmosphere. This is complicated by the fact that the atmospheric density and vehicle aerodynamic properties are not sufficiently well known and because entry conditions cannot be precisely controlled. Thus, on-board guidance, navigation, and control systems must be able to determine in real time the true flight environment and adapt quickly to that environment. In most aerocapture applications, the required guidance and navigation computations cannot be performed on the ground and then relayed to the vehicles because the round-trip transmission time is too great or because of communications blackout problems.

Controlled flight through Earth's atmosphere at hypersonic speeds has been successfully performed by the Apollo and Space Shuttle vehicles. However, the objective in each case has been to land safely on Earth, not to fly an atmospheric skip trajectory. This difference in terminal objective necessitates a significant difference in guidance logic for most aerobrake applications.[23] AFE studies have investigated and proposed designs for navigation and control systems to meet the requirements for lunar return. However, these studies have not addressed high entry velocities (14 km/sec or greater) and different vehicle configurations with L/D between 0.5 and 1.5, nor have they addressed the problems associated with aerobraking at a distant planet, where different and relatively uncertain atmospheric properties will be encountered.

A typical planetary aerocapture scenario is illustrated in Fig. 14. The entry corridor is limited by the overshoot boundary, or skip-out, region at the top and the undershoot boundary, or reentry, region at the bottom. At the present time, a 1° corridor width at Mars is the design guide. The exit trajectory must be precisely controlled. The vehicle must emerge from the atmosphere with the correct six-dimensional state vector so as to be able to transfer, with a minimum propellant cost, into the desired final orbit. Merely emerging from the atmosphere with the correct velocity magnitude is generally not sufficient. In surface landing applications, the three-dimensional position must be accurately controlled with small velocity components at surface touchdown. Trajectory selection must also consider vehicle heating rates, g-loads, and, in the case of Mars, high-elevation topographic features.

Essentially all applications of hypersonic flight high in a planetary atmosphere, whether for the purpose of aerocapture, performing a synergetic plane change, or descending to the surface from orbit require accurate knowledge and control of the vehicle's position and velocity at the time of atmospheric entry, as well as on-board computation of the vehicle's position and velocity while in the atmosphere. Autonomous on-board GN&C systems are generally needed to meet these requirements.[24] The presence of density waves in the Martian atmosphere coupled with the uncertainties in vehicle aerodynamic characteristics at hypersonic speeds greatly

complicate the problem. Consequently, it is important that on-board guidance schemes be either insensitive to deviations in atmospheric and aerodynamic properties from their nominal values or else adaptable to the real environment based on a real-time, on-board determination of that environment. In addition, the guidance schemes, must be able to compensate for deviations from the flight-path angle and for other position and velocity components at the time of entry.

There are several significant challenges in GN&C which must be addressed for future space exploration. However, aerocapture GN&C does not appear to present any technological issues as serious as those described above for aerothermodynamics, TPS materials, and vehicle design.

Summary

The critical technologies for aerobrake development have been reviewed. The state of the art of each of four disciplines—vehicle concepts and configuration design, aerothermodynamics, thermal protection system materials, and guidance, navigation, and control—is described, as well as future requirements for space exploration. Critical developments in the areas of aerothermodynamics and TPS materials are required to meet the flight-readiness levels for proposed missions to the moon and Mars. The vehicle configurations for specific missions must await the results of in-depth trade-off studies to optimize a variety of factors. Guidance, navigation, and control systems require additional work but appear to be in excellent shape to meet the necessary requirements of the exploration program.

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Table 1: Aerobrake mass fractions for lunar missions

	MDSSC ASTV	JSC ASTV	LaRC ASTV	GD ASTV	MMC LTV
Weights (MT) / Mission	GEO	GEO	GEO	GEO	LUNAR
Vehicle dry weight	3.96	5.43	5.25	4.48	15.66
Return payload weight	0	6.80	5.44	0.91	0
Residuals (est)	0.35	0.92	0.62	0.43	1.50
Total return weight	4.31	13.15	11.31	5.92	17.16
Aerobrake weight	0.81	0.84	1.75	1.30	1.81
A/B structural eff. %	18.80	6.40	15.50	22.00	10.50

Table 2: Aerobrake parameters for Mars missions

Mission	AFE	Lunar return	Mars return (reusable)	Mars return (capsule)	Mars entry (samp. ret.)	Mars entry (manned)
Entry speed (km/sec)	10	11	12.5	14	6–8	7–9
Decel. altitude (km)	75	75	66	65	40	40
Stag. press. (atm)	0.03	0.05	0.2	0.3	0.1–0.15	0.1–0.2
Max. heating (w/cm^2) conv./rad.	40/7	50/30*	150/500*	500/900*	100/10*	100/10*(L/D≈1) 30/50*("0.3)
TPS	Non-ablat.	Non-ablat. or ablat.	Ablating	Ablating	Ablating & non-ablat.	Non-ablat. or ablat.
% Ionization	0.5–1	5–10	15	25	0–1	0–2
Non-equil. Δrelax/Δstandoff	0.4	0.2	0.15	0.1	Large	Large
Maj. of bound layer	Lam.	Lam.	Turb.	Lam./Turb.	Turb.	Turb.
Important physics						
Rad./abl. interact	None	Small	Large	Large	Small	Small
Chem./catalicity	Yes	Yes	No	No	Yes	Yes
Bound. layer tran.	No	No	Yes	Yes	Yes	Yes
Aerodynamics (slip, abl., etc.)	Yes	Yes	Maybe	Maybe	Maybe	Maybe

*Based on equilibrium theory

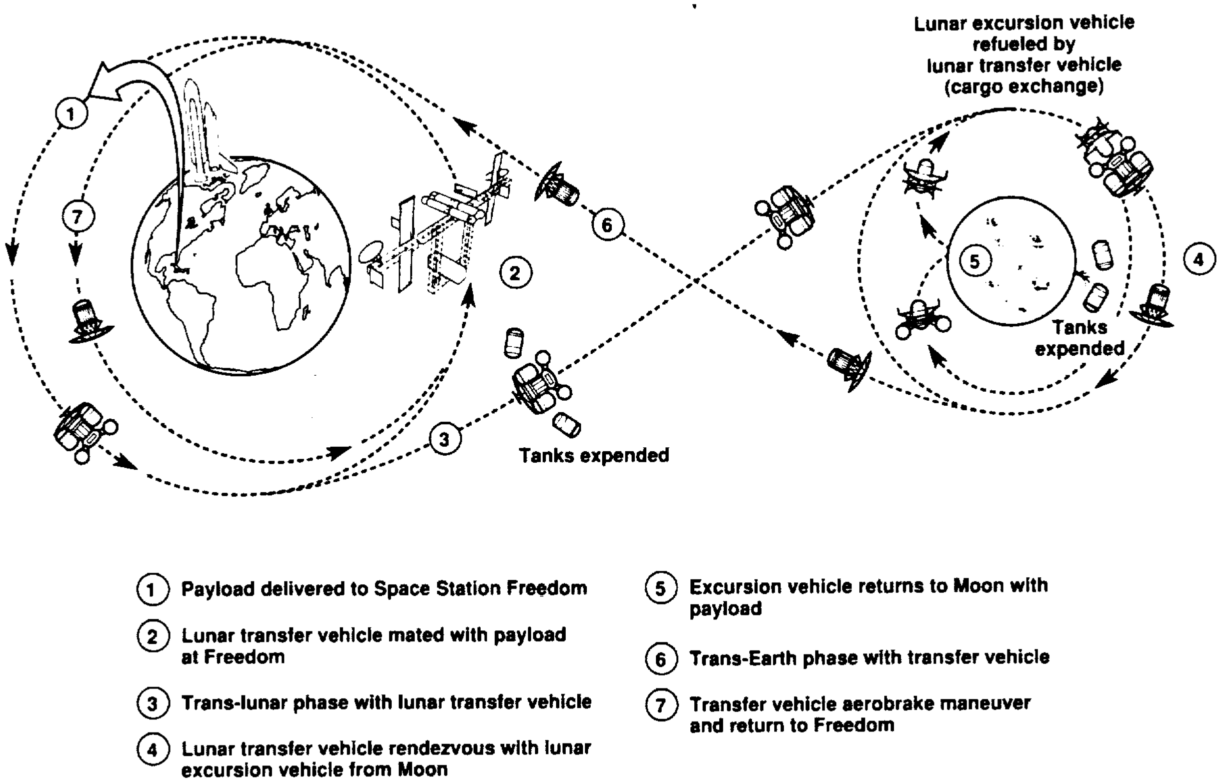


Fig. 1 Lunar mission profile (from Ref. 12).

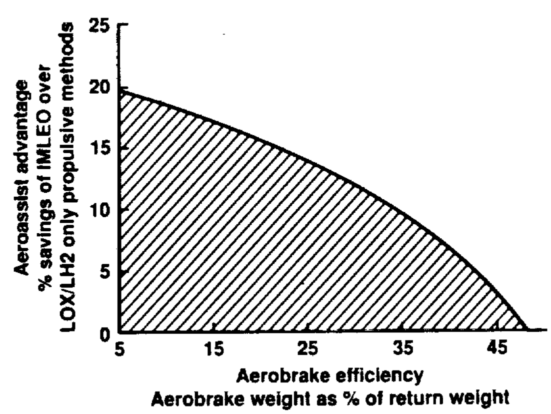


Fig. 2 Lunar mission aeroassist advantage.

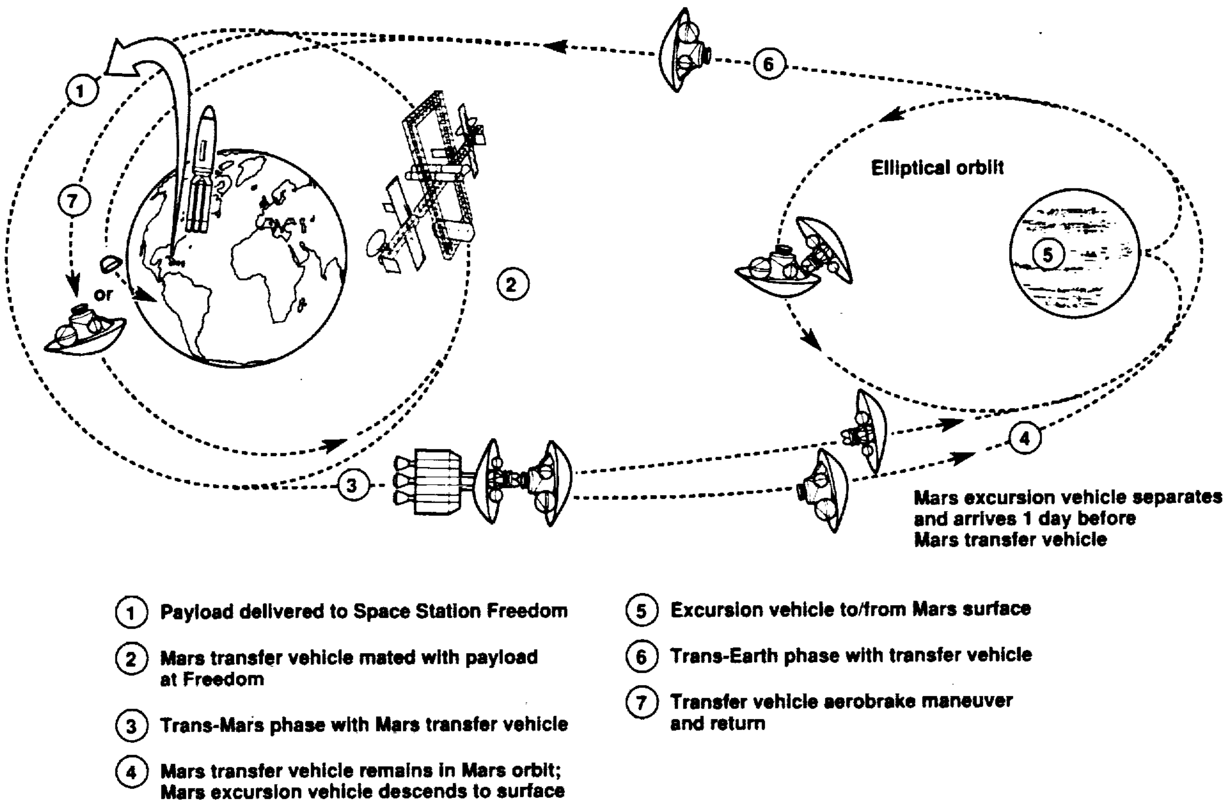


Fig. 3 Mars mission profile (from Ref. 12).

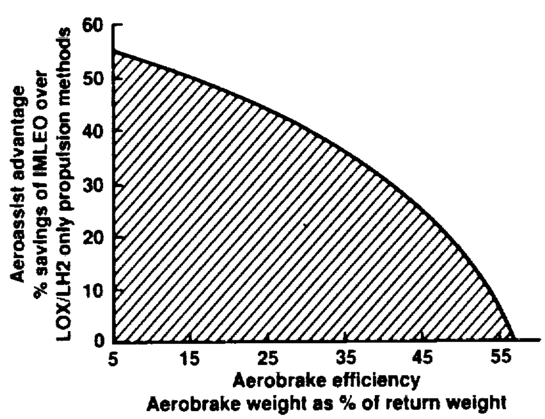


Fig. 4 Mars mission aeroassist advantage (from Ref. 5).

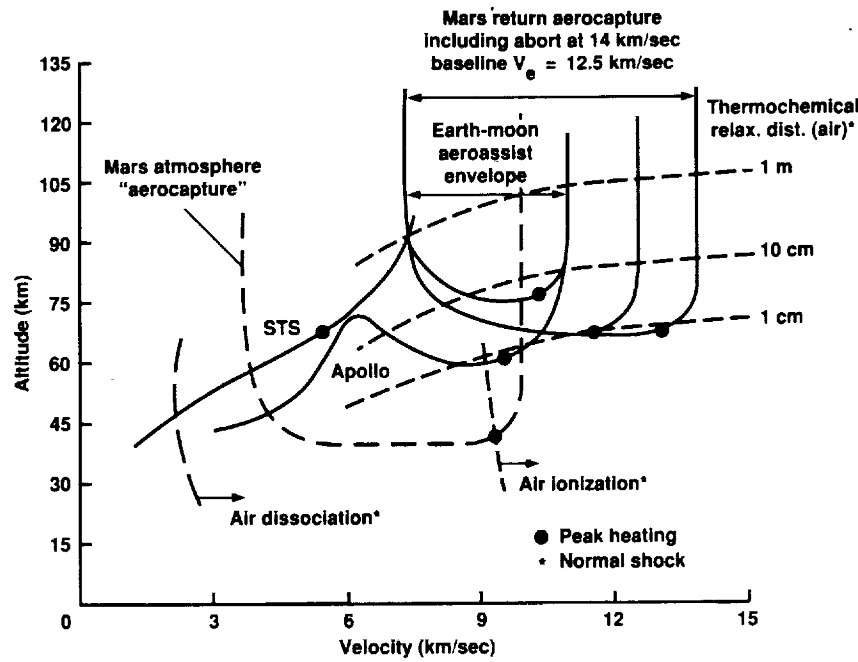


Fig. 5 Aerobrake/aerocapture flight regimes.

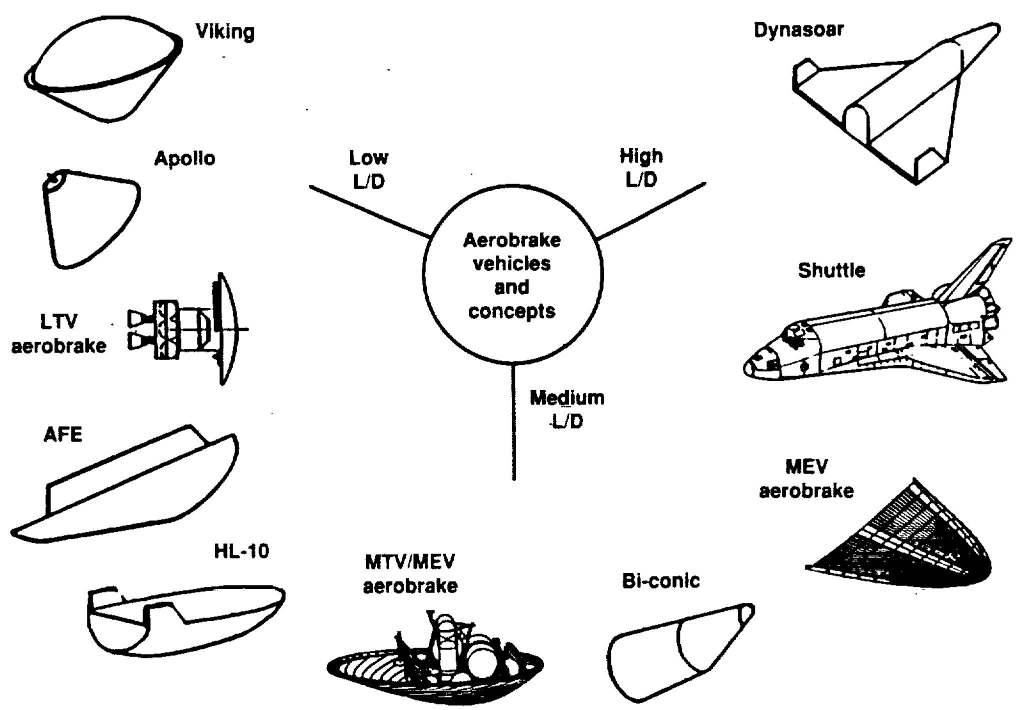


Fig. 6 Aerobrake vehicles and concepts.

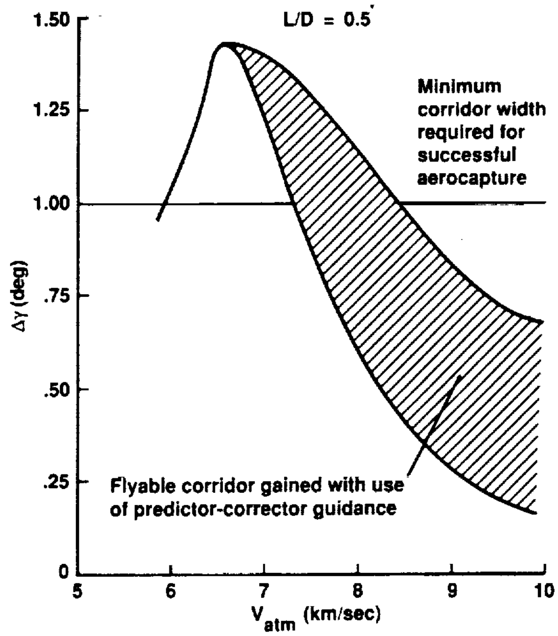


Fig. 7 Mars flyable corridor width (from Ref. 16).

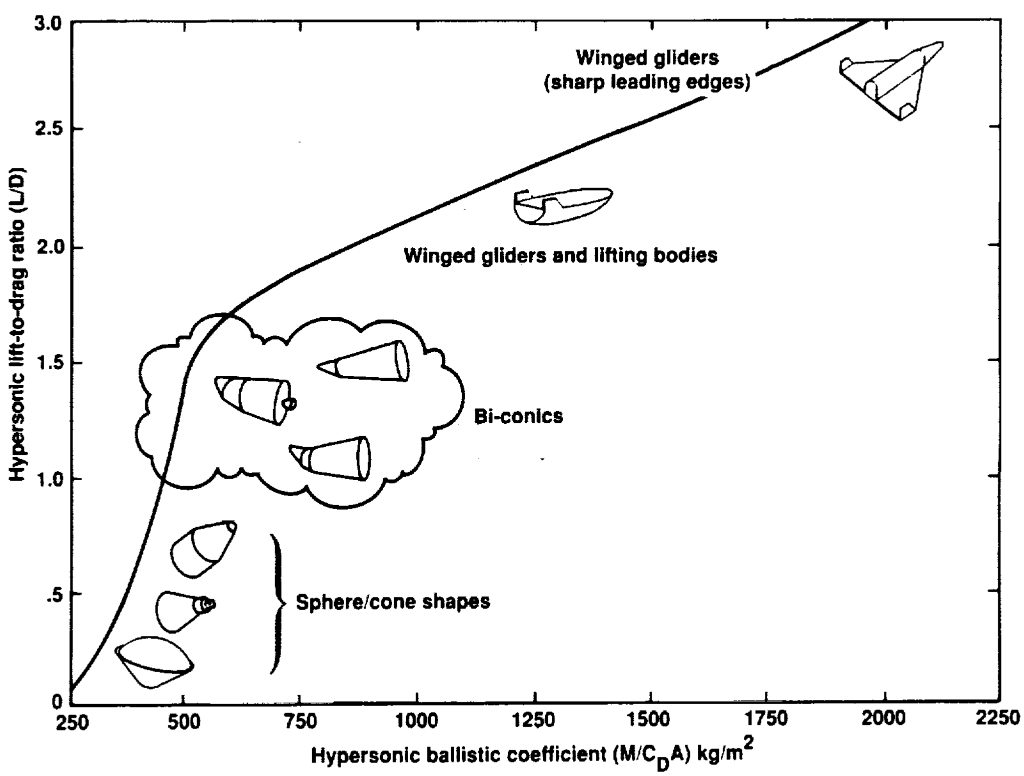
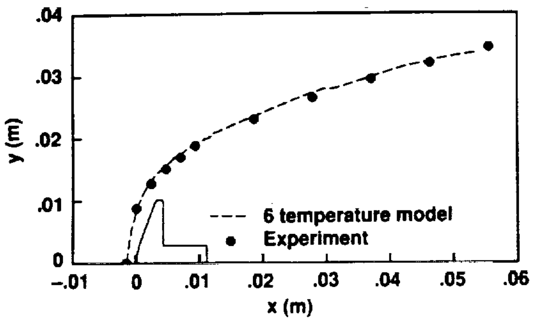
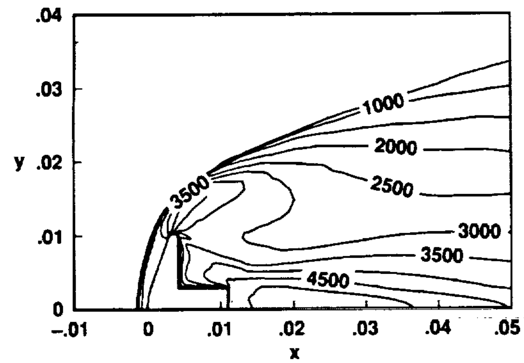


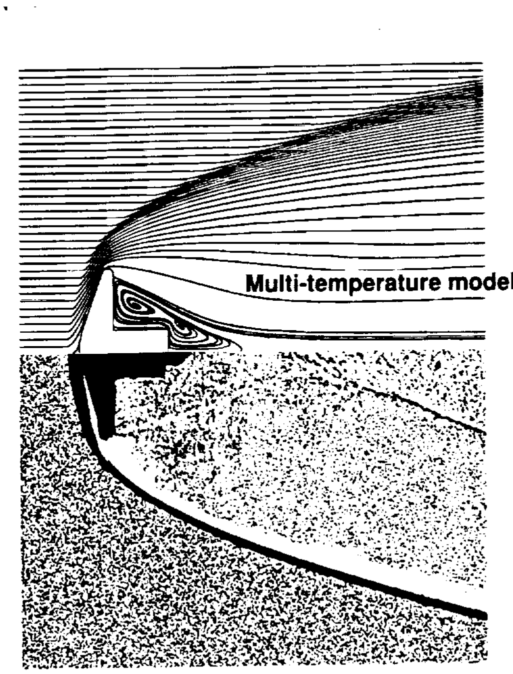
Fig. 8 Potential aerodynamic shapes for aerocapture vehicles: m = 5,000 kg



Shock detachment on axisymmetric AOTV model in air



Translational-rotational temperature contours for axisymmetric AOTV model (K)



Comparison between computed streamlines and experimental shadowgraph on axisymmetric AOTV model in air

Fig. 9 Two-dimensional axisymmetric multi-temperature Navier-Stokes hypersonic flow simulation (from Ref. 18).

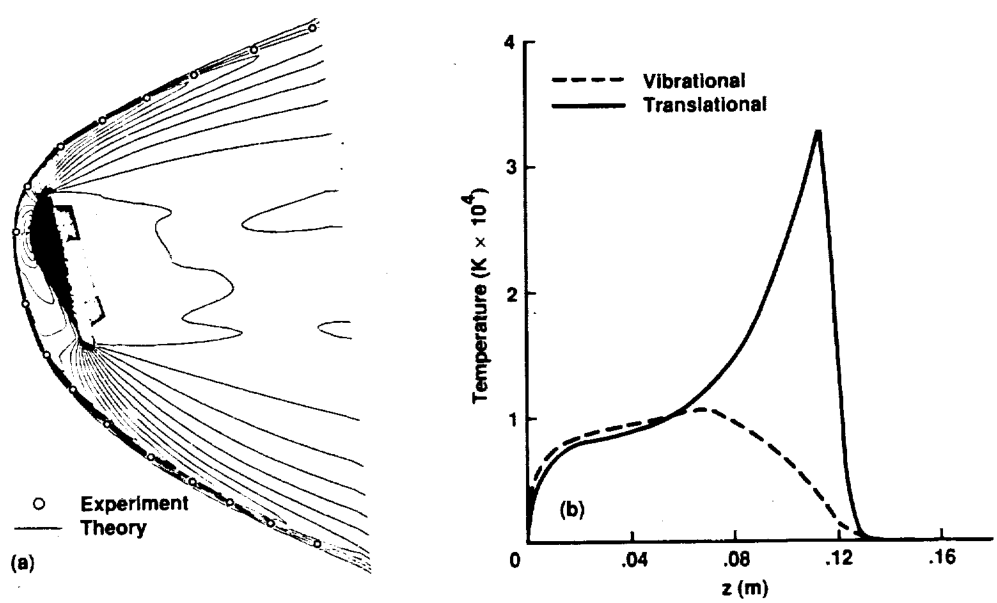


Fig. 10 Thermochemical nonequilibrium flow around the AFE vehicle: Mach No. = 5.0. (a) Experimental and calculated shock shapes; (b) comparison of vibrational and translational stagnation-line temperature profiles (from Ref. 19).

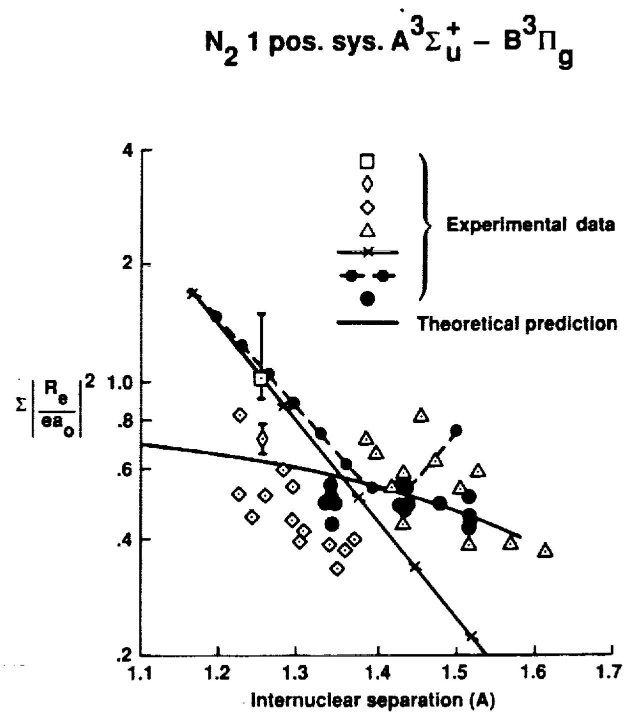


Fig. 11 Electron transition moments for the N_2 first positive system.

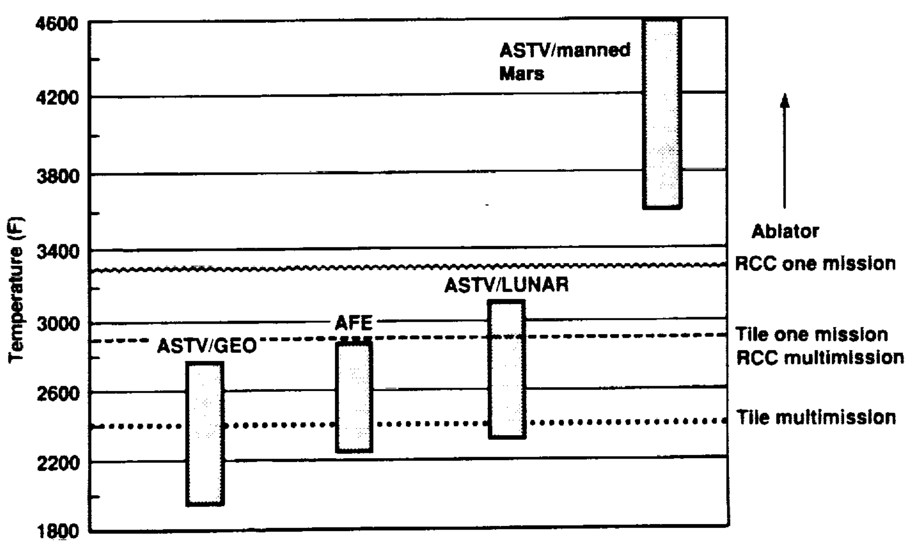


Fig. 12 Thermal protection system radiation equilibrium temperatures.

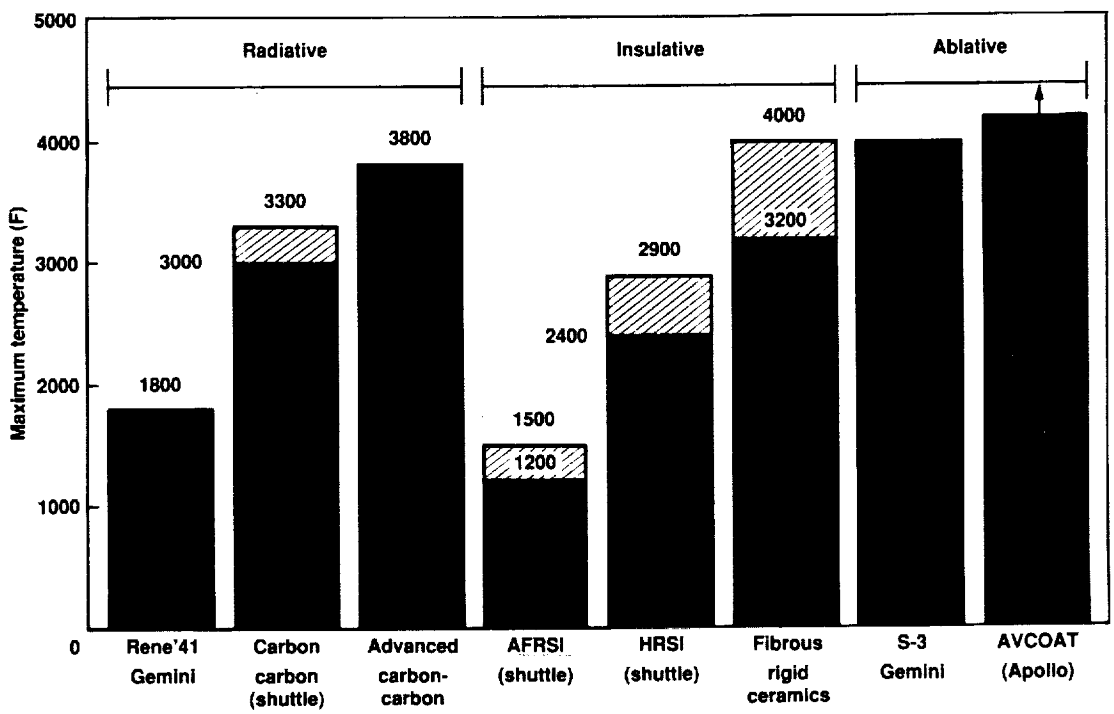


Fig. 13 Candidate thermal protection material capabilities.

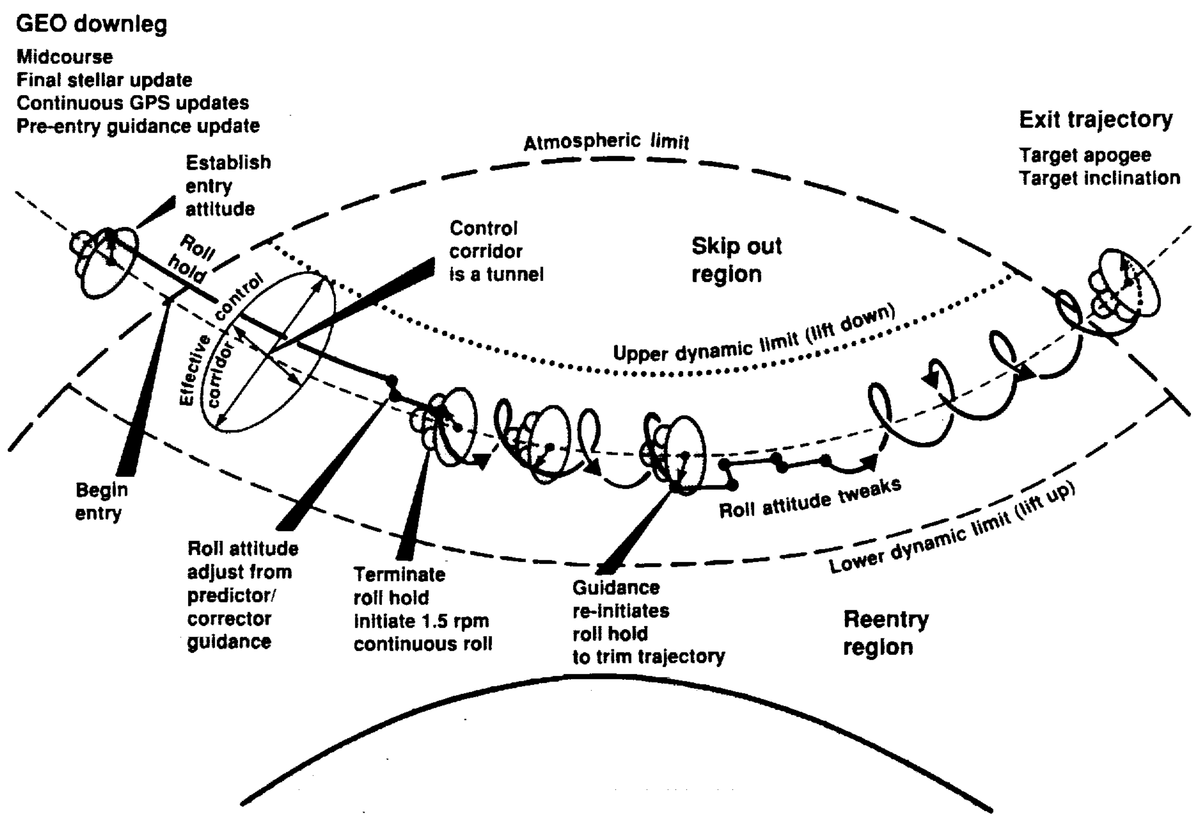


Fig. 14 Planetary aerocapture overview.

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15. Supplementary Notes

Point of Contact: David M. Cooper, Ames Research Center, MS 258-5, Moffett Field, CA 94035-1000, (415) 604-4500 or FTS 464-4500

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16. Abstract

Aerobraking is one of the largest contributors to making both lunar and Mars missions affordable. The use of aerobraking/aeroassist over all-propulsive approaches saves as much as 60% of the initial mass required in low-Earth orbit (LEO), thus reducing the number and size of Earth-to-orbit launch vehicles. Lunar transfer vehicles (LTV), which will be used to transport personnel and materials from LEO to lunar outposts, will aerobrake into Earth's atmosphere at approximately 11 km/sec on return from the lunar surface. Current plans for both manned and robotic missions to Mars use aerocapture during arrival at Mars and at Earth return. At Mars, the entry velocities will range from about 6 to 9.5 km/sec, and at Earth the return velocity will be about 12.5 to 14 km/sec. These entry velocities depend on trajectories, flight dates, and mission scenarios and bound the range of velocities required for the current studies. In order to successfully design aerobrakes to withstand the aerodynamic forces and heating associated with these entry velocities, as well as to make them efficient, several critical technologies must be developed. These are vehicle concepts and configurations, aerothermodynamics, thermal protection system materials, and guidance, navigation, and control systems. This paper describes the status of each of these technologies and outlines what must be accomplished in each area to meet the requirements of the Space Exploration Initiative.

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